BEHAVIOR ANALYSIS FOR A SUBASSEMBLY IN ALEO SATELLITE

Sorin DRĂGHICI[1], Horia-Alexandru PETRESCU[2], RalucaVOICU[3], Anton HADĂR[4]

This paper aims to design a better component hosting a satellite electronics in order of reduce weight without changing the overall dimensions and initial materials. The authors have proposed two new variants which were analyzed by the finite element method to determine the behavior of assemblies subjected to temperature variations that occur in space.

Keywords: design, aluminum, satellites, space thermal cycles

1. Introduction

Space industry is one of the most important in present being the provider of everything related to modern lifestyle: data transmission, weather forecast, GPS etc. Although the benefits of this industry are huge, so are the costs to enable this technology, the cost of sending a kilogram to a low earth orbit LEO can reach up to $ 20,000 [1]. Therefore, the main objective of the research related to this area is reducing the weight, especially weight savings component that generates itself after a series of lower costs (transport, handling, and easier structure).

The objective of this paper is to optimize the design of an electronic housing in a satellite in order to reduce weight and improve mechanical performance.

For a component to be viable in space, it must meet several conditions, the most important being:

- Vibration behavior (the first natural frequency for this subassembly must be under 150 Hz) ;
- Structural strength (the stresses must be in the elastic domain) ;
- Resistance to extreme temperatures;

[1]As., Dept. of Strength of Materials, University POLITEHNICA of Bucharest, Romania, e-mail: sdrg79@yahoo.com
[2]Lect., Dept.of Strength of Materials, University POLITEHNICA of Bucharest, Romania, e-mail: horia.petrescu@upb.ro
[3]Eng., CompositeMaterials Laboratory, INCDT COMOTI, Bucharest, Romania, e-mail: raluca.voicu@comoti.ro
[4]Prof., Department of Strength of Materials, University POLITEHNICA of Bucharest, Romania, e-mail: anton.hadar@upb.ro

- Radiation protection for interior components. [2]

This study aims to validate that only by choosing another subassembly configuration, one can improve its behavior. The reference electronic housing was implemented in a satellite named “Proba 2”which was launched by the European Space Agency on November 2, 2009 [3]. In this paper, comparative studies were made between the proposed geometries and the existing geometry using the finite element method. It should be mentioned that all three structures are made of aluminum.

2. Presentation of a subassembly geometric design

The performance of a component can be described by an equation of the form [4]:

$$p = f(F, G, M) \tag{1}$$

in which p is performance, F- functional requirements, G-geometric parameters and M-material properties.

Functional requirements are specified by the constructive role. Geometric parameters derive from the specified geometry and boundary conditions. Functional requirements, geometrical parameters and material properties are generally separable. In this paper, the functional requirements and the materials used for the aluminum boxes remain the same, so the only way in which the improvement of subassembly behavior can be possible is by changing the geometrical parameters.

The shape and dimensions of the electronic housing containing the advanced data and power management system (ADPMS) as well as the construction requirements are imposed by its position within the satellite. Therefore, to meet its purpose, the overall dimensions of the subassembly remain the same: 460 mm x 154 mm x 250 mm, with a wall thickness of 2 mm. For the same reasons, the outlets which engage electronic cards weren't changed from one embodiment to another.

As a point of reference the existing Proba 2satellite'selectronic housing was adopted, as described in [4]. The configuration shown in fig. 1was obtained.

It was considered that a box made by fewer components generates a multitude of benefits which are directly reflected in the total cost of the mission. These are:

a) fewer molds needed for the components;
b) fewer undertakings;
c) increased rigidity;
d) reduced weight.

Therefore, we proposed two variants of the electronic housingthat consist of two components. These are shown schematically in Figures 2 and 3, [5].

After the design of all the components a weight evaluation has been made of each box and the results are given in Table 1. It should be mentioned that all the components of the box were made of aluminum, and the supports for the electronic plates placed inside the box have been made of a composite glass / epoxy (FR4) used mostly in manufacturing electronic boards.

Table 1

Mass of existing and new electronic housing

Model	Mass [kg]
Existent	7,3287
Variant 1	7,6021
Variant 2	7,3171

The new geometry has to meet the same general conditions of strength and working as a real case which houses the electronics. These are detailed in Table 2.

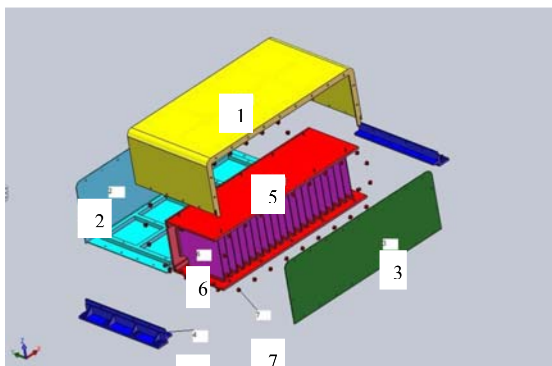

Fig.1. Components for existing electronic housing on Proba2:
1- hat section; 2- base and rear panel; 3- front panel; 4- mounting rails; 5- wedge locks; 6- electronic plates; 7- aluminum rivets;

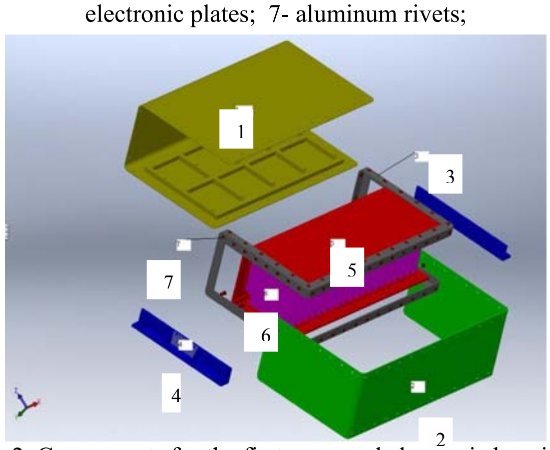

Fig.2. Components for the first proposed electronic housing:
1- upper and lower panels; 2- front and lateral panel; 3- aluminum rail; 4- mounting rails; 5- wedge locks; 6-electronic plates; 7- aluminum rivets;

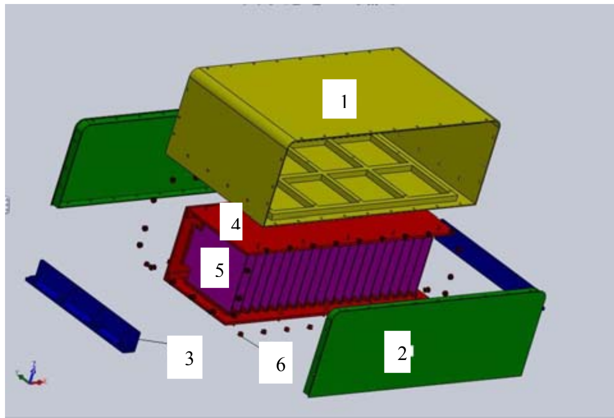

Fig. 3 Components for the second proposed electronic housing:1- upper and lower panels; 2- lateral panel; 3- mounting rails; 4- wedge locks; 5-electronic plates; 6- aluminum rivets;

Table2

General conditions

Demands	Aluminum
Dimension	460 mm x 154 mm x 250 mm
Wall thickness	2 mm
Weight	standard
Access	standard
Structural strength	standard*

* the stresses must be in the elastic domain

3. Finite elements simulations for the considered models

For all simulations, the following assumptions were made:

a) all the boxes are made of the same aluminum alloy (see Table 3);

b) all electronics, including outlets that are installed are made of composite material (FR4, see Table 3);

c) all degrees of freedom on the underside were suppressed on the side brackets that attach the satellite box;

d) Contact problems that arise between parts are ignored and these were considered as bonded.

Table3

Materialsused in the finite element analyses

Material	E_x [MPa]	E_y [MPa]	υ_{xy}	υ_{yz}	ρ [kg/m^3]	α [K^{-1}]
Aluminum alloy	71000	71000	0.33	0.33	2770	$22.2\cdot10^{-6}$
Glass/epoxy FR4	24000	21000	0.136	0.118	1850	$25\cdot10^{-6}$

The satellites in LEO type orbit (450 -1200 km) and high elliptical orbit may be exposed to external disturbances as: infrared radiation of the earth, the sun and Albedo radiation. The sun and Albedo radiation can be neglected for large elliptical orbits [6]. Incident maximum flux commonly occurring for an orbit of 550 km for a flat surface normal to nadir (the point of the celestial sphere that is directly opposite the zenith and vertically downward from the observer) are q_{IR} ~ 200 W/m^2, $q_{ALB, MAX}$ ~ 410 W/m^2 (medium value on orbit is <150 W/m^2). Time eclipse can reach half hour for circular orbit and several hours for elliptical orbits [6].

These flows can be translated in thermal operating mode for satellites which can range from a low of -100 °C to + 150 °C maximum, depending on its orbit.

Orbiting the earth's shadow or on their own shade can be a challenge in maintaining the structural stability of the satellite and its functionality. The satellite can quickly switch on the cold side (usually -125 °C) when it is in the shade. Alternatively, it may be subjected to extremely high temperatures (typically + 150 °C) when exposed to sunlight. In addition, there is a substantial amount of heat generated by the equipment on board. Therefore, the materials used to build the satellite must be highly adaptable in order to protect from extreme heat and cold. Multilayer insulation blankets (MLI) are ideal for protecting sensitive tools in these extreme conditions [7].

Multilayer insulation materials consist of light reflective films assembled in several thin layers. These layers are typically made of polyamide and/or polyester films (depending on the design, it may be from 5 to 30 layers) are deposited by evaporation on aluminum (99.99%), on one or both sides. Multilayer films helps in managing heat by reducing the emissivity of each layer [7].

Since it is very difficult to design an insulation blanket MLI reflecting 100% of incident radiation, a multilayer film design can vary from a few simple layers to a comprehensive range of blankets that completely surrounds many of the external satellite's components. By design these layers usually reflect back 95% of the radiation from the satellite. The total web when the radiation energy passes through its thickness is effectively a 100% reflective barrier.

Ideally the temperatures necessary for the satellite to operate at full capacity are between -10°C and + 40 °C. This is achieved by using insulating materials that will work MLI passive and active systems. Passive systems are best described as materials selected during the design process, such as MLI blankets, which provide heat and radiation barrier to delay and control the flow of energy.

Active systems can be best described as energy sources such as battery powered heaters who are working to keep the environment at the target temperature (+ 10°C) during cold cycles. Conduction lines can be made of flexible copper wire with adjustable geometry. Heat pipes are other means for very effective thermal conductivity. They are mostly represented by nickel and titanium alloys in bar form annular groove extruded. The typical profile has a diameter of 8 mm and thermal resistance less than 0.1 K/W. The liquid commonly used as a transfer medium is ammonia,[6].

With these active and passive internal protection, the components of a satellite are not subjected to sudden changes of temperature. Temperature does not vary between extreme limits existing in LEO conditions (see Table 4) [9].

Table 4

Thermal regimes of space satellites components [6, 7, 8]

Component	Operational temp. [^{o}C]	Survival temp. [^{o}C]
Digital electronics	-10 ÷ +40	-20 ÷ +70
Analogs electronics	0 ÷ +40	-20 ÷ +70
Batteries	10 ÷ 20	0 ÷ +35
Propulsion system	10 ÷ 50	0 ÷ +60
Solar panels	-100 ÷ +125	-100 ÷ +125

Considering all these factors and the fact that the electronic unit is housing the satellite's electronics (which should be operational) it can be concluded that it will not be exposed to different thermal cycles than electronics operating margin, therefore finite element analyses were made taking into account a thermal cycle between -10 and +40 °C [9] which occurs during the time satellite is orbiting earth. The time needed for Proba 2 to make a full revolution around the earth is 100 minutes. Therefore, the thermal cycles applied varied with 0.5^{o}C/minute. The thermal loads were applied on the outer contour of models.

The number of nodes and elements for each analyzed configuration is shown in Table 5. For all models the used element type was SOLID 187.

Table 5

Nodes and elements for finite elements analyses

Model	Number of nodes	Number of contact elements	Total number of elements
Existent	183434	59522	142281
Variant 1	216792	81338	164573
Variant 2	221813	50008	157405

In the performed analyses the following values of the stresses and safety factors were obtained(the safety factor (S.F.) is defined as the ratio between failure stress and maximum stress). Thus, in Figure 4, the von Mises equivalent stress for the initial geometry is depicted. The highest value for stress (93.378 MPa) is located at the edge of a mounting circular hole. This value is considered to be a local effect of the stress state. The average stress value obtained in the model is close to a value of 31 MPa, which is spread throughout the entire geometrical model.

Fig. 5 presents the von Mises equivalent stress first proposed subassembly. The highest value for stress (76.081 MPa) is obtained at the same location as in the previous model (initial geometry), thus confirming the local effect of stress near the mounting hole. This smaller value is a consequence of the smaller values for average stress (25 MPa) obtained throughout the first proposed subassembly. The von Mises equivalent stress values for the last proposed subassembly are presented in Figure 6. The maximum stress is 75.042 MPa and represents a local effect obtained at the same location as in for the previous models. This last case has the smallest value of stress at the concentrator due to the better distribution of the average stress throughout the model, a smaller value for average stress (16 MPa)being obtained. In Figure 7, the safety factor for the initial geometry is depicted. The minimum value (0.76) is located at the edge of a mounting circular hole. This value is considered to be a local effect of the stress state. The average S.F. value obtained in the model is close to a value of 5.5.

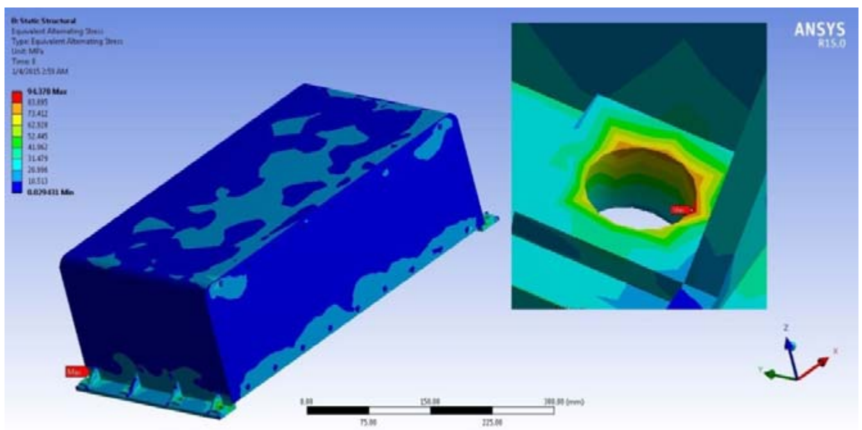

Fig. 4 Stresses obtained for existing subassembly

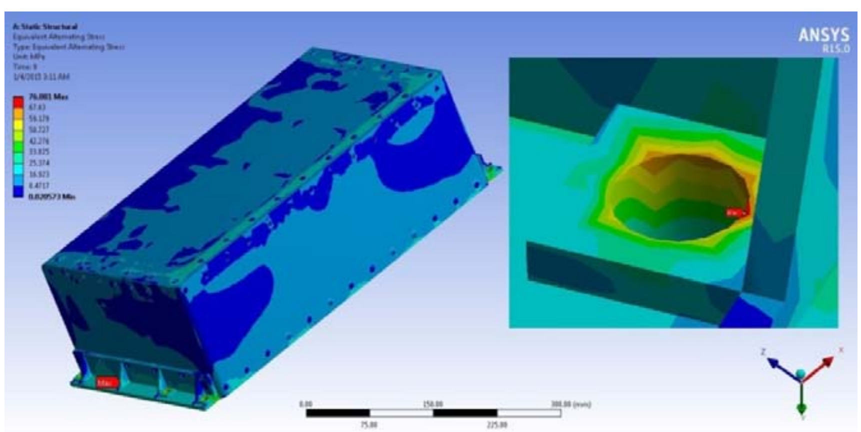

Fig. 5 Stresses obtained for first proposed subassembly

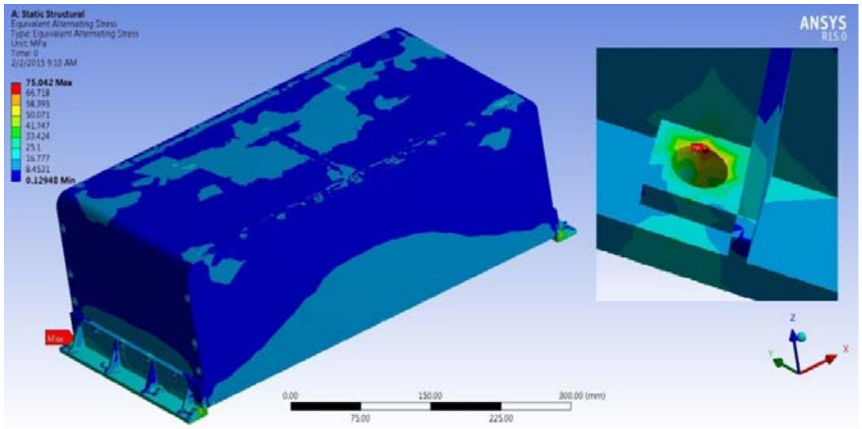

Fig. 6 Stresses obtained for second proposed subassembly

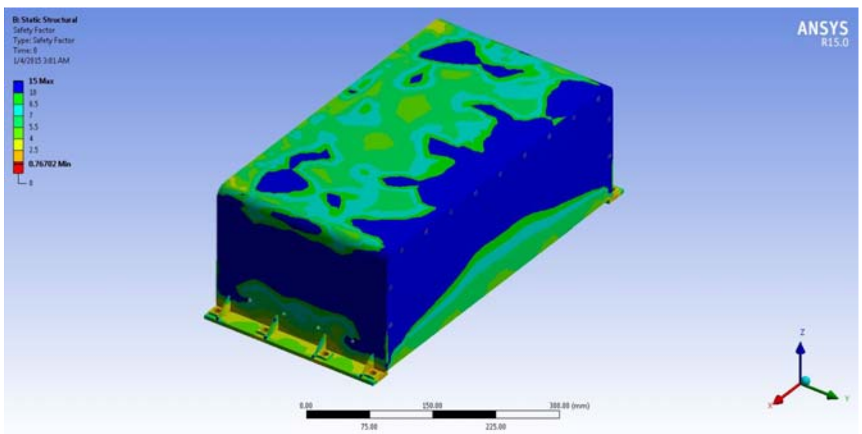

Fig. 7 Safety factor for existing electronic housing.

Fig. 8 presents safety factor for the first proposed subassembly. The lowest value for S.F. (1.08) is obtained at the same location as in the previous model (initial geometry), thus confirming the local effect of stress near the mounting hole. The average S.F. value obtained in the model is 5.

In space industry the weight is a decisive element for the design stage. Therefore, the safety factor used in the design process is between 1.10÷ 1.25.

The safety factor values for the last proposed subassembly are presented in Fig. 9. The minimum safety factor is 1.10 and represents a local effect obtained at the same location as for the previous models. But, in this case the safety factor in above the minimum imposed value.

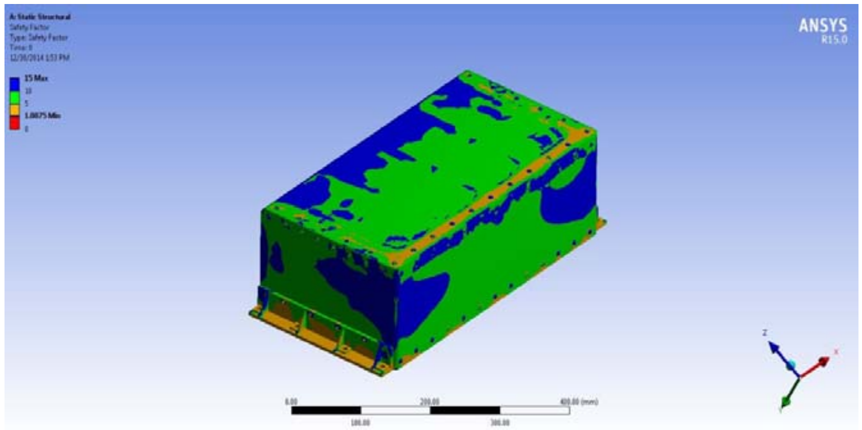

Fig. 8 Safety factor for the first proposed electronic housing.

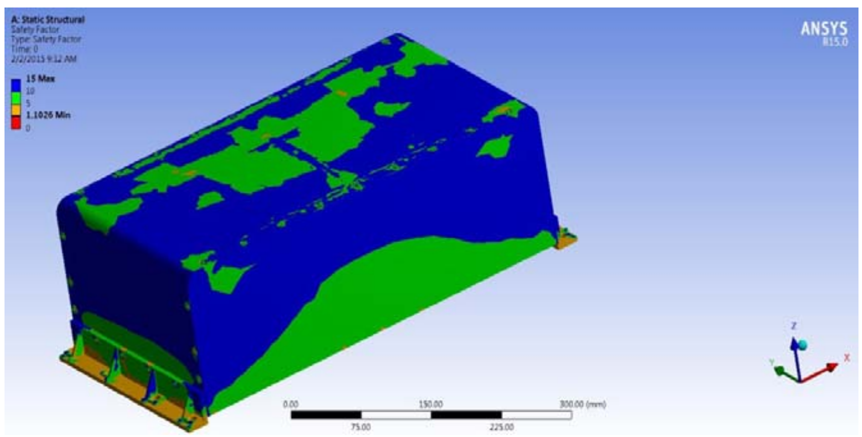

Fig. 9 Safety factor for the second proposed electronic housing.

4. Conclusions

After the finite element analyses were made, the following conclusions can be drawn:

- All variants show maximum stresses on the outer brackets;
- Both proposed variants exhibit lower stresses than the existing subassembly, the maximum value of the von Mises stress decreasing with 20.5%;
- Both proposed alternatives present a safety factor greater than unity unlike the existing version, with a safety factor of 0.76, case in which the structural integrity of the assembly is jeopardized;
- The second proposed subassembly achieved both goals presenting a decrease in mass by about 3% and an improved performance. Since this cannot be considered a significant achievement in terms of weight, further studies are to be

conducted in order to diminish the wall thickness, and, consequently, to decrease the weight until the entire strength capacity of the box is reached. Also, the use of different lighter materials can be further considered.

- The stresses on the electronic housing wall are smaller than the yield limit of the material and, therefore, do not affect the structural integrity, as it is also indicated by the obtained safety factors. For example the 6061-T6 aluminum alloy which was used for these simulations has an ultimate tensile strength of at least 300 MPa and yield limit of at least 241 MPa. Even if in certain points of the structure the stresses would be greater than the yield limit, the structural integrity is not compromised (if higher stresses don't result consistently over large regions). But, if the stresses are smaller than the yield limit, the safety of the box is increased and the possibility of failure due to an accidental load during moving on the orbit decreases.

It can be concluded that a judicious design can bring significant savings to the mission, especially since the design can be further modified regarding the wall thickness of the box until it reaches the maximum allowable strength.

Acknowledgement

The work has been funded by the Sectoral Operational Programme Human Resources Development 2007-2013 of the Ministry of European Funds through the Financial Agreement POSDRU/159/1.5/S/134398

R E F E R E N C E S

[1]. Web 15Nov 2013, http://www.nasa.gov/centers/marshall/news/background/facts/astp.html_%20prt.htm

[2] Payload Flight Equipment Requirements and Guidelines for Safety–Critical Structures National Aeronautics and Space Administration International Space Station Program Johnson Space Center Houston, Texas, 2002

[3] Web 15 Nov 2013, https://directory.eoportal.org/web/eoportal/satellite-missions/p/proba-2

[4] *F. Garcia and K. Kurvinen* , (2006) Radiation Shielding Study of Advanced Data and Power Management Systems (ADPMS) Housing Using Geant4, IEEE Nuclear Science Symposium Conference.

[5] *S. Drăghici, F. Baciu, R.Voicu, A. Hadăr* (2014) Modeling the behavior at launching for a satellite's subassembly, Conference Proceedings of the Academy of Romanian Scientists, Productica, 39-50

[6] *V. Baturkin* "Micro-satellites thermal control-concepts and components."ActaAstronautica**56**.1 (2005): pp. 161-170.

[7] Dunmore Web 27 Dec 2014 http://www.dunmore.com/products/multi-layer-films.html

[8] *Ph. Poinas*, Satellite thermal control engineering - presentation for SME 2004, http://www.tak2000.com/data/Satellite_TC.pdf

[9] *J.M. Defise* et. al, SWAP, a novel EUV telescope for Space Weather, Proceedings of SPIE - The International Society for Optical Engineering 09/2007; DOI: 10.1117/12.731784